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RESEARCH MEMORANDUM

for the

U. S. Air Force

PRELIMINARY WIND-TUNNEL INVESTIGATION OF THE PERFORMANCE
OF REPUBLIC F-105 WING-ROOT INLET CONFIGURATIONS AT
VARIOUS ANGLES OF ATTACK AND A MACH NUMBER OF 2.01

Coord. No. AF-163

By Walter L. Kouyoumjian

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NATIONAL ADVISORY COMMITTEE
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RESEARCH MEMORANDUM

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PRELIMINARY WIND-TUNNEL INVESTIGATION OF THE PERFORMANCE
OF REPUBLIC F-105 WING-ROOT INLET CONFIGURATIONS AT
VARIOUS ANGLES OF ATTACK AND A MACH NUMBER OF 2.01

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SUMMARY

A 1/13-scale model of the forebody of the Republic F-105 with twin-duct wing-root inlets was tested in the Langley 4- by 4-foot supersonic pressure tunnel through a range of angle of attack from -4° to 15° at a Mach number of 2.01 and a Reynolds number of approximately 3.4×10^6 per foot.

The tests were made with four configurations which incorporated varying amounts of sweep and stagger of the inlet leading edges, modifications to the areas of the boundary-layer diverter floor plate, and modifications to the area of the boundary-layer diverter bleed slots. The highest overall pressure recovery at an angle of attack of 0° (average total-pressure recovery, 0.84; mass-flow ratio, 0.98) was achieved with a configuration having an inlet leading-edge sweep angle of 58° with no leading-edge stagger. Stagger was found to improve the angle-of-attack performance, but at a sacrifice in inlet efficiency for an angle of attack of 0° . The boundary-layer diverter floor height, of the order of one boundary-layer thickness, was satisfactory for bypassing the fuselage boundary layer. The boundary-layer diverter-plate bleed slots were effective in increasing the total-pressure recovery of the inlet. The total-pressure-recovery contour plots, taken at the compressor-face station, indicate the existence of high-velocity "cores" throughout the inlet operating range.

INTRODUCTION

At the request of the U. S. Air Force, a twin-duct wing-root inlet incorporating a boundary-layer diverter and slotted inlet floor (at the inlet throat section) was tested in the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 2.01 and a Reynolds number of 3.4×10^6 per foot. This supersonic inlet was a 1/13-scale model of the forebody of the Republic F-105. Although the inlet model included provisions for varying the throat area and also for bypassing air in each of the twin ducts (compressor-face bleed), the inlet was tested as a fixed-geometry inlet in the present investigation.

In the investigation four major configurations were used, with each configuration employing different inlet-lip sweep angle and lip leading-edge stagger angle. The performance of the boundary-layer diverter and the longitudinal slots in the inlet throat were investigated, and the effect of modifying these components was observed. The inlet tests were performed at an angle of sideslip of 0° throughout a range of angle of attack from -4° to 15° . The data are presented with a limited analysis.

SYMBOLS

H	average total pressure, lb/sq ft
H_L	local total pressure, lb/sq ft
H_0	free-stream total pressure, lb/sq ft
M	Mach number
m	inlet mass flow, slugs/sec
m_0	free-stream mass flow, slugs/sec
$\frac{A_{min}}{A_0}$	contraction ratio; inlet minimum area to projected inlet frontal area
$\frac{H}{H_0}$	average total-pressure recovery at compressor face
$\frac{H_L}{H_0}$	local total-pressure recovery at compressor face

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$\frac{m}{m_0}$ mass-flow ratio, based on inlet capture area of 0.0343 sq ft

α angle of attack, deg

APPARATUS

Tunnel

The tests were performed in the Langley 4- by 4-foot supersonic pressure tunnel at a free-stream Mach number of 2.01, tunnel stagnation pressure of approximately 1 atmosphere, and stagnation temperature of about 100° F. The Reynolds number of the tests was approximately 3.4×10^6 per foot. The pressure data were recorded photographically from an inclined manometer board using mercury as the working fluid, with the board inclined at an angle of 30° with the horizontal. The tunnel schlieren apparatus had a continuous light source for observations during runs and employed a spark discharge to make instantaneous photographs of the supersonic flow field.

Model

The model was constructed of metal and was supplied by the Republic Aviation Corporation. The model included the fuselage forebody of the F-105 and stub wings in which the wing-root inlets were mounted (figs. 1 and 2). The inlet incorporated variable-throat area (a translating plug in the supersonic diffuser shown in fig. 3) and had provisions for mass-flow bypass in each subsonic duct section. Neither of these provisions were utilized in the present tests, however. No instrumentation was provided for making force measurements. Inlet mass flow was measured by using a calibrated flat-plate orifice located downstream of the total-pressure rake station and upstream of the throttle. The model was mounted in the tunnel from the rear by a steel sting support. The supersonic diffuser had an initial compression of 5° followed by isentropic compression of 13°. The contraction ratio $\frac{A_{min}}{A_0}$ was 0.67. The inlet lip

section was removable so that the angle of sweep could be varied. (See fig. 3.) The boundary-layer diverter was set for all tests at a height approximately equal to the boundary-layer thickness of 0.25 inch. (See fig. 1.)

The boundary-layer diverter had eight longitudinal slots (0.046 inch by 0.52 inch) at the inlet throat section (figs. 3 and 4). The chamber below the slots was exhausted to the free stream on either side of the

inlet (see fig. 3). The floor area was varied by increasing the external dimensions by 0.5 inch. (See fig. 3.) A boundary-layer transition strip, consisting of a 0.5-inch strip of no. 60 carborundum grains in shellac, was utilized at several locations on the fuselage forebody to assure turbulent boundary-layer flow into the inlet.

The inlet axis was canted down -2.5° relative to the fuselage center line.

The sketch of figure 1 shows the general configuration of the model and indicates the junction of the individual ducts at the compressor-face station. The total-pressure rake at the compressor-face station was utilized to give local total-pressure recoveries and average-pressure recoveries at this station.

Configuration 1.- Configuration 1 employed an inlet leading-edge sweep of approximately 58° normal to the fuselage center line (fig. 3). The intersection of the inlet lip and the diverter floor was slotted to provide a starting spillage area. This configuration was investigated with the original diverter floor plate (fig. 3) and with longitudinal slots in the diverter plate both opened and closed. This configuration was tested with the boundary-layer transition strip located at station 20.2 and with the strip removed.

Configuration 2.- Configuration 2 employed inlet leading-edge stagger; that is, the top of the inlet lip had a sweep of 58° while the bottom had a sweep of approximately 65° (fig. 3). The diverter floor slots were open, and the boundary-layer transition strip was positioned $1\frac{1}{2}$ inches in front of the inlet. This configuration was investigated with two different diverter floor plates: the original one, and one in which all external dimensions were increased by 0.5 inch (see fig. 3). For a subsequent test the floor slots on the larger boundary-layer plate were enlarged to 0.07 inch (0.07 inch by 0.52 inch).

Configuration 3.- For configuration 3 the upper leading-edge sweep was eliminated so that the upper lip was normal to the direction of the airstream, while the lower lip was still swept at 65° (fig. 3). This configuration gave the maximum allowable inlet leading-edge stagger available with the present model.

The boundary-layer diverter and the diverter floor plate were a modification of configuration 2 (fig. 3).

Configuration 4.- For configuration 4 both upper and lower inlet leading edges were swept back to an angle of approximately 65° . The modified boundary-layer diverter of configuration 2 was used in this configuration (fig. 3).

RESULTS

The results of the present series of tests are given in graphic form as average compressor-face total-pressure recovery $\frac{H}{H_0}$ plotted against mass-flow ratio $\frac{m}{m_0}$ for various angles of attack (figs. 5 and 6). The compressor-face total-pressure recovery profiles are plotted as contour plots in figures 7 to 10. These profiles are faired as two separate half sections because of the vertical divider that extends to a location immediately before the compressor-face station.

Schlieren photographs of the external flow field are presented in figures 11 to 13. The test points, for which contour plots and schlieren photographs are shown, were chosen as representative of subcritical and supercritical mass-flow ratios.

DISCUSSION

Configuration 1

From figure 5(a) it is observed that the peak pressure recovery of the inlet of configuration 1 at an angle of attack of 0° was 0.84. The value was decreased by 0.05 when the boundary-layer diverter plate slots were closed. The presence of a boundary-layer transition strip did not affect the peak pressure recovery of the inlet. Figure 5(a) also shows an unexplained dip in the inlet operating curve, which was reproduced in two separate runs. (Note point with and without boundary-layer transition strip.)

The schlieren photographs of figure 11 indicate that the boundary-layer diverter height is adequate with respect to the fuselage boundary-layer thickness.

The total-pressure-recovery contour plots of configuration 1 (fig. 7) indicate that for supercritical inlet mass-flow ratios the air flow tends to form high-velocity "cores," introducing both radial and circumferential air-flow distortion at the compressor-face station. The range of Mach numbers calculated for supercritical flows such as $\frac{m}{m_0} = 1.02$ (fig. 7) is from approximately 0.25, corresponding to the lowest pressure recoveries, to approximately 0.55 for the highest pressure recoveries. The total-pressure-recovery contour plots are plotted as two separate halves because of the presence of a vertical divider between the two inlets. The presence of this vertical divider is believed to introduce a "dead"

region; however, in the present tests no instrumentation was introduced to determine the air flow behind this partition.

The stable operating range for this configuration was quite limited, and the inlet became unstable (one duct became subcritical) at very high subcritical mass-flow ratios.

Configuration 2

For configuration 2 the inclusion of inlet leading-edge stagger serves to shift the peak pressure recovery $\left(\frac{H}{H_0} = 0.85, \frac{m}{m_0} = 0.90\right)$ from an angle of attack of 0° to an angle of attack of 4° (fig. 5(b)) and serves to increase the inlet performance and maximum supercritical mass flow rate. The effect of increasing the boundary-layer diverter-plate area is to decrease the peak pressure recovery at $\alpha = 4^\circ$ by 0.015 (fig. 5(b)). Additional boundary-layer bleed area (obtained by increasing the width of the boundary-layer diverter-plate slots) had no noticeable effect on the peak pressure recoveries obtained. The total-pressure-recovery contour plots (fig. 8) indicate a similar pattern of high-velocity cores as was presented for configuration 1. The calculated Mach number variations were of the same order as was calculated for configuration 1 (0.25 to 0.55).

The schlieren photographs for this configuration (fig. 12) do not indicate any appreciable differences from those for configuration 1 (fig. 11). An additional photograph of the modification using the larger boundary-layer diverter floor plate is included for reference.

Configuration 3

No schlieren photographs were made for configuration 3. Only the curves of pressure recovery against mass-flow ratio (fig. 6(a)) and the total-pressure-recovery contour plots (fig. 9) are shown. The peak pressure recovery for this configuration was 0.87 at $\alpha = 8^\circ$ and $\frac{m}{m_0} = 0.67$. From figure 6(a) it is noted that the maximum mass-flow ratio is reduced, in comparison with configuration 1 (fig. 5), and the peak of the pressure-recovery curve has been shifted to $\alpha = 8^\circ$.

The calculated compressor-face Mach numbers for this configuration indicate a decrease in the range of Mach number, with a Mach number of 0.25 corresponding to the lowest pressure recovery and a Mach number of 0.45 corresponding to the highest.

Configuration 4

From figure 6(b) it is seen that the maximum mass-flow ratio of configuration 4 has decreased by 0.1 in comparison with configuration 1. The peak pressure recovery for this configuration was 0.83 for $\frac{m}{m_0} = 0.78$ at $\alpha = 0^\circ$. Figure 6(b) indicates that the inlet operating curves for $\alpha = 0^\circ$ and $\alpha = 4^\circ$ are superimposed. It is believed that the increased spillage of the configuration is responsible for the apparent insensitivity to small angles of attack.

The curves of figure 6(b) were obtained from tests in which the inlet mass-flow ratio was reduced from the maximum (supercritical) value to the mass-flow ratio at which one or both inlet ducts become unstable. Although investigations were made throughout this stable operating range for all the previous configurations, the inlet geometry precluded schlieren observation inasmuch as the inlet flow field was obscured by the external leading edge (fig. 3, configuration 3); therefore, the exact value of the lower mass-flow ratio point was dubious. It is assumed that the characteristics displayed by configuration 4 are typical of the limited stable operating range of the model.

The schlieren photographs of figure 13 indicate a strong shock system standing ahead of the inlet minimum section. It is believed that the position of these shocks is the reason for the increased spillage and the decreased maximum inlet mass-flow ratio and the decreased total-pressure recovery.

The total-pressure-recovery contour plots (fig. 10) indicate the existence of high-velocity "cores" of the order of approximately $M = 0.53$ corresponding to the highest pressure recoveries and with Mach numbers of approximately 0.30 corresponding to the lowest pressure recoveries. It is further noted that as the inlet mass-flow ratio is decreased, one inlet duct becomes unstable while the other inlet duct is still in the high subcritical range. As previously stated, this condition is considered to be the lower limit for inlet operation.

SUMMARY OF RESULTS

A 1/13-scale model of the Republic F-105, with twin-duct wing-root inlets, was tested at the Langley 4- by 4-foot supersonic pressure tunnel at a free-stream Mach number of 2.01 and a Reynolds number of approximately 3.4×10^6 per foot. The following results were noted:

1. The highest overall pressure recovery obtained at an angle of attack of 0° is 0.84 for a mass-flow ratio of 0.98 for a configuration

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incorporating boundary-layer diverter slots, no inlet leading-edge stagger, and inlet leading edges swept at an angle of 58° .

2. The investigation also disclosed that inlet leading-edge stagger improves angle-of-attack performance and increases the maximum supercritical mass-flow ratio of the inlet at angles of attack.

3. The boundary-layer diverter floor slots are effective in increasing the inlet total-pressure recovery. The boundary-layer diverter floor height, of the order of one boundary-layer thickness, is satisfactory for bypassing the fuselage boundary layer.

4. Analysis of the compressor-face total-pressure-recovery contour plots indicates the presence of both circumferential and radial air-flow distortions for all the points taken for configurations 1 and 2 (maximum mass-flow-ratio configurations). From analysis of the compressor-face total-pressure-recovery contour plots and schlieren photographs, in some cases, it is observed that the two individual ducts of the system act unsymmetrically as mass-flow ratio is reduced and model angle of attack is varied.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., November 28, 1956.

Walter L. Kouyoumjian

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Approved:

John V. Becker

John V. Becker
Chief of Compressibility Research Division

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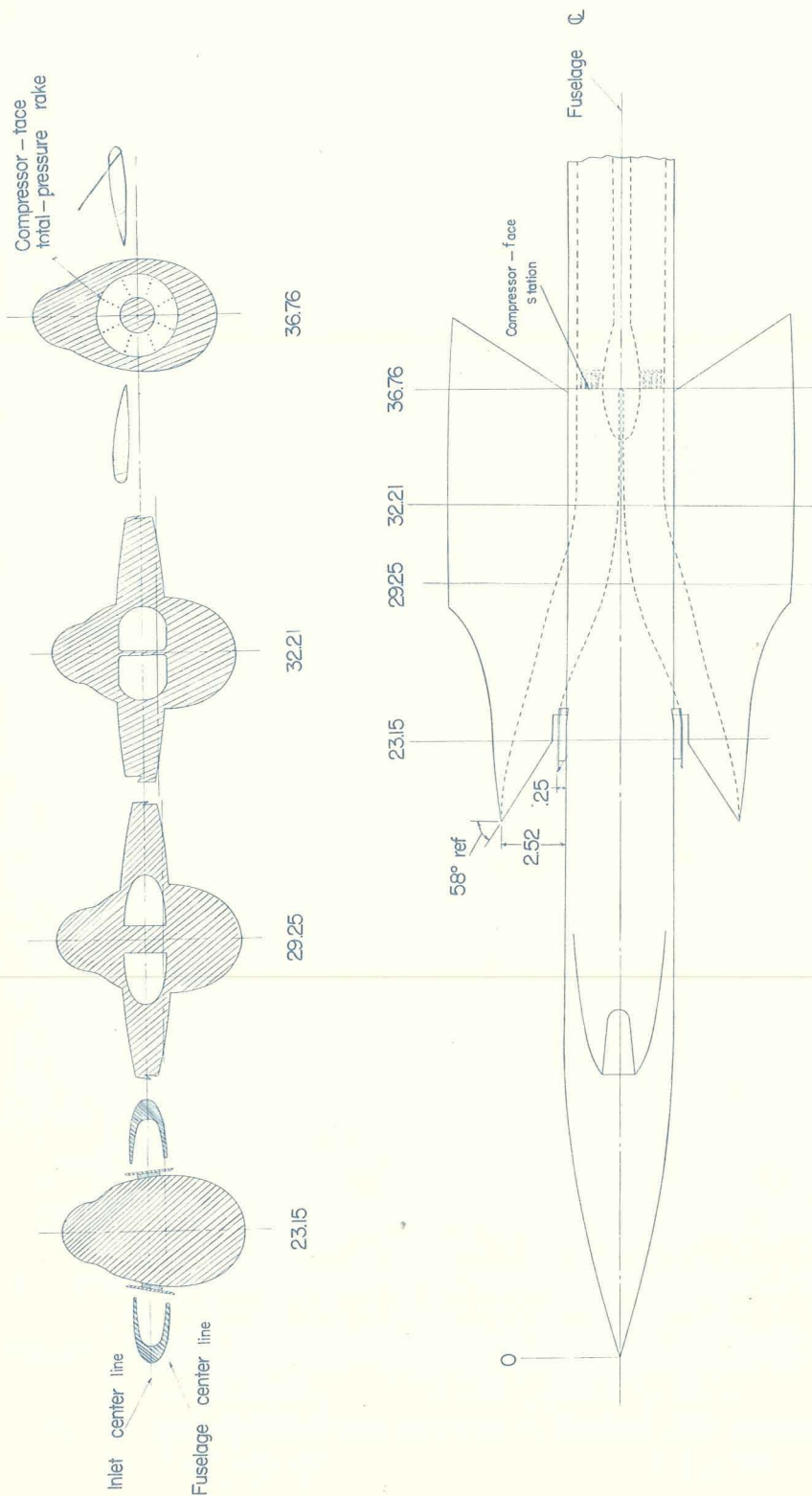
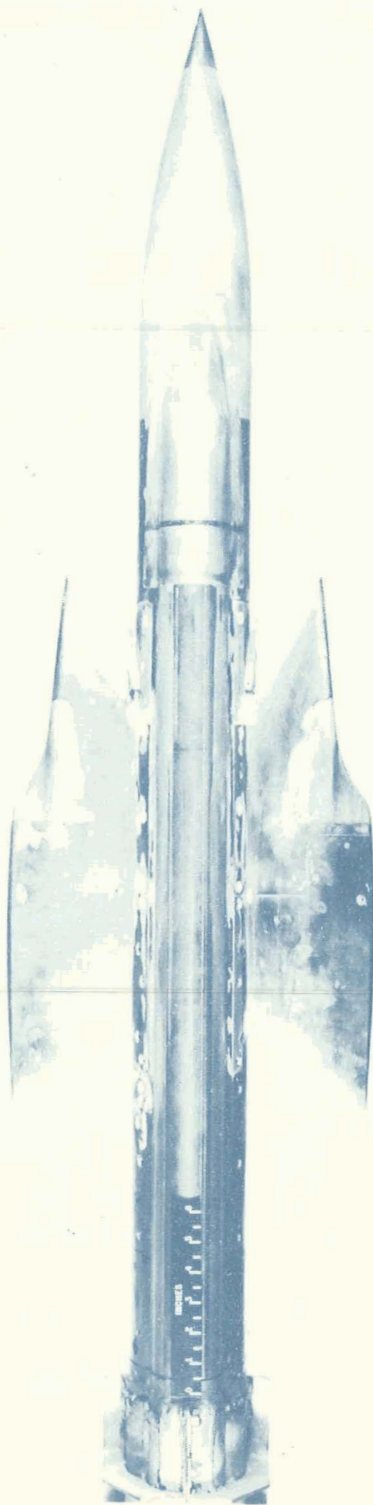
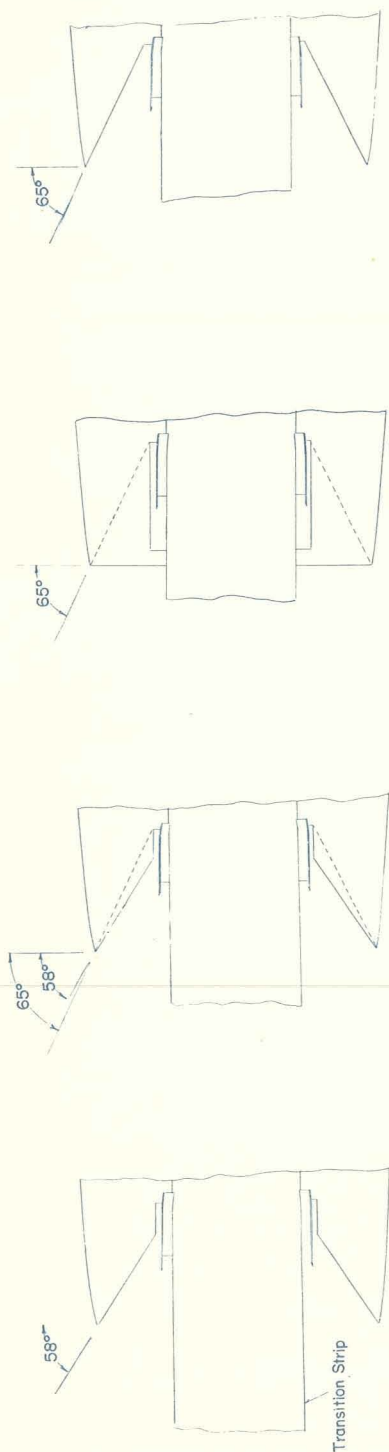


Figure 1.- Layout of model. All dimensions in inches.



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Figure 2.- Top view of model (configuration 4).

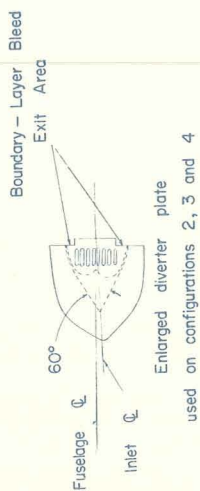


Configuration No. 4

Configuration No. 3

Configuration No. 2

Configuration No. 1



Enlarged diverter plate

used on configurations 2, 3 and 4



Original diverter plate
used on configurations 1 and 2

Boundary – Layer Diverter Plate

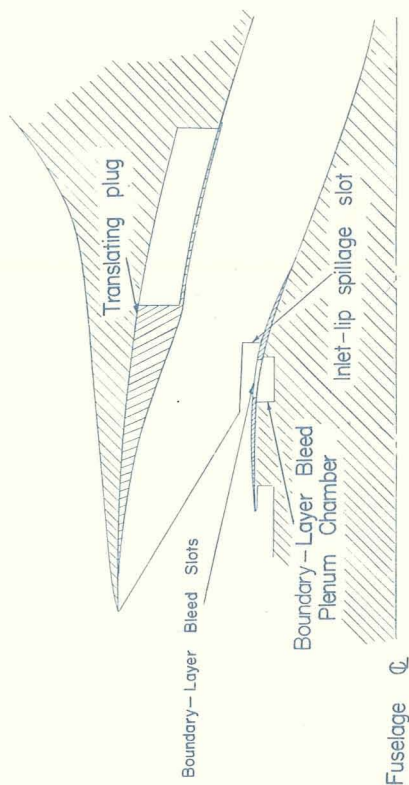
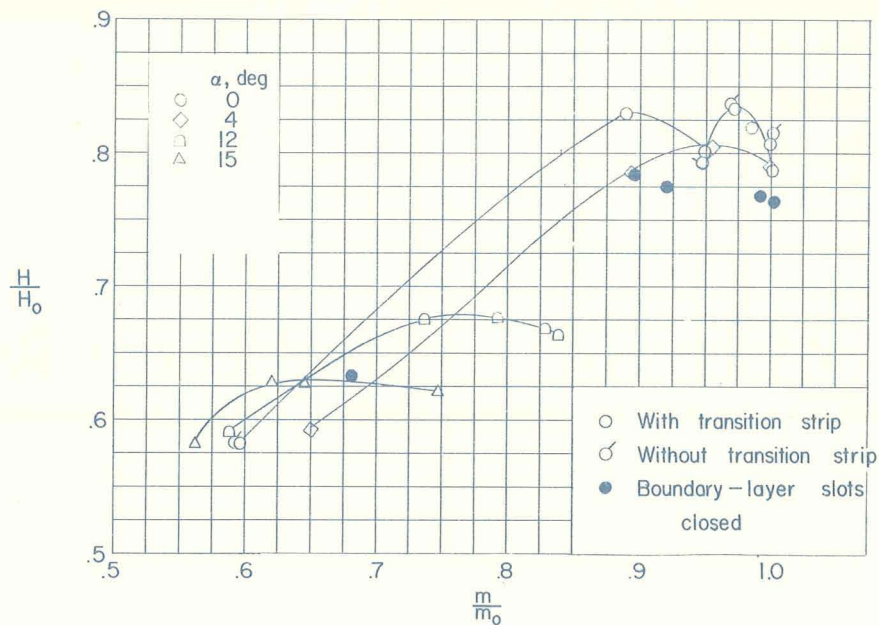


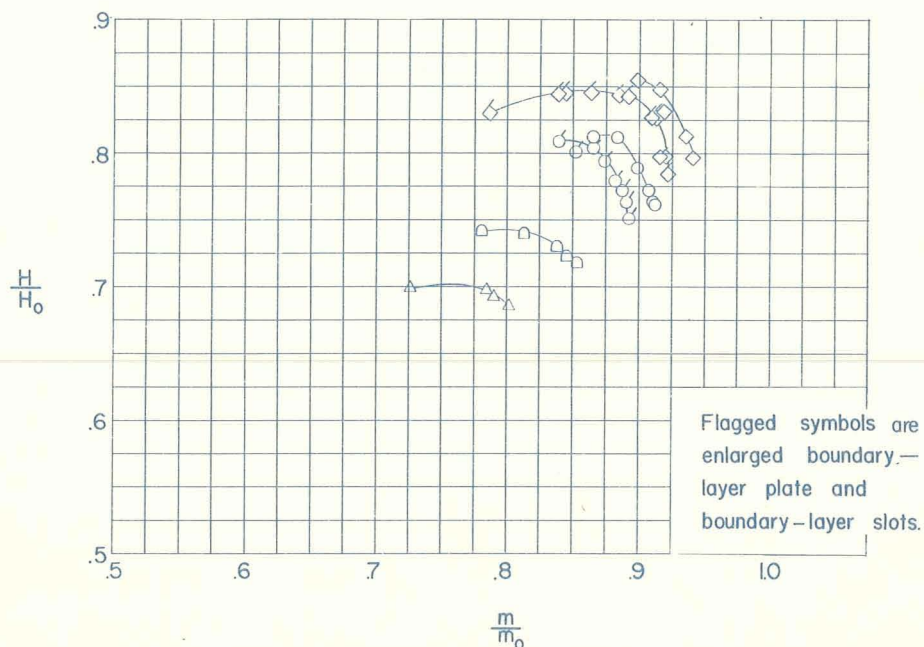
Figure 3.- Details of inlet configurations. All dimensions are in inches.



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Figure 4.- Photograph of inlet model showing large boundary-layer plate with slotted boundary-layer bleed at inlet throat.

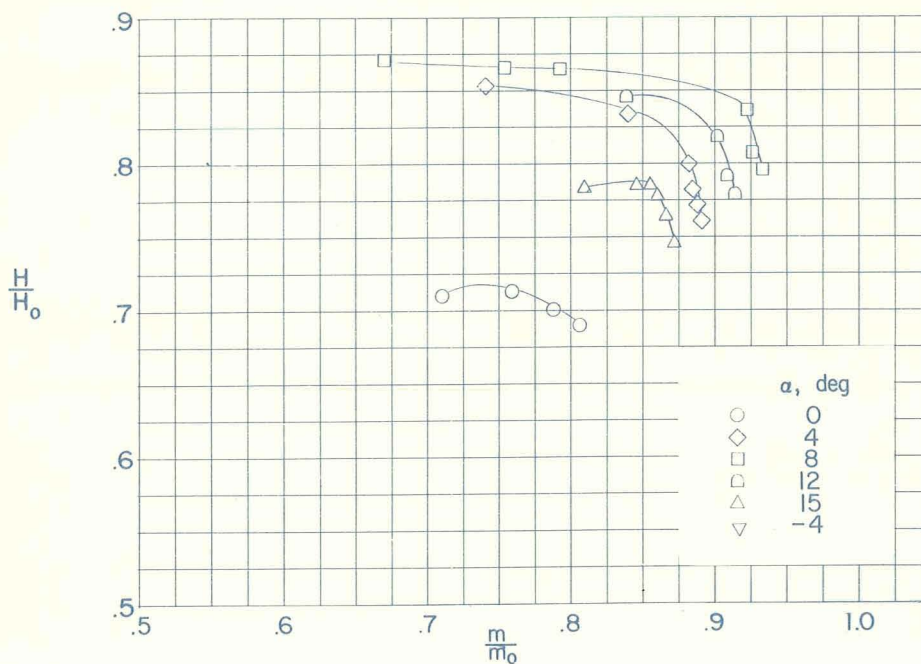


(a) Configuration 1.

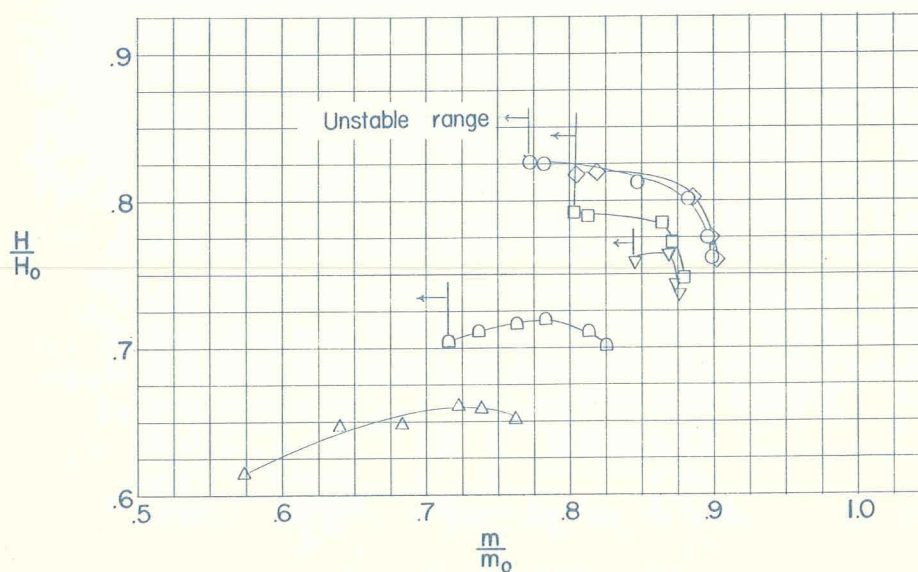


(b) Configuration 2.

Figure 5.- Average pressure recovery against mass-flow ratio for configurations 1 and 2.



(a) Configuration 3.



(b) Configuration 4.

Figure 6.- Average pressure recovery against mass-flow ratio for configurations 3 and 4.

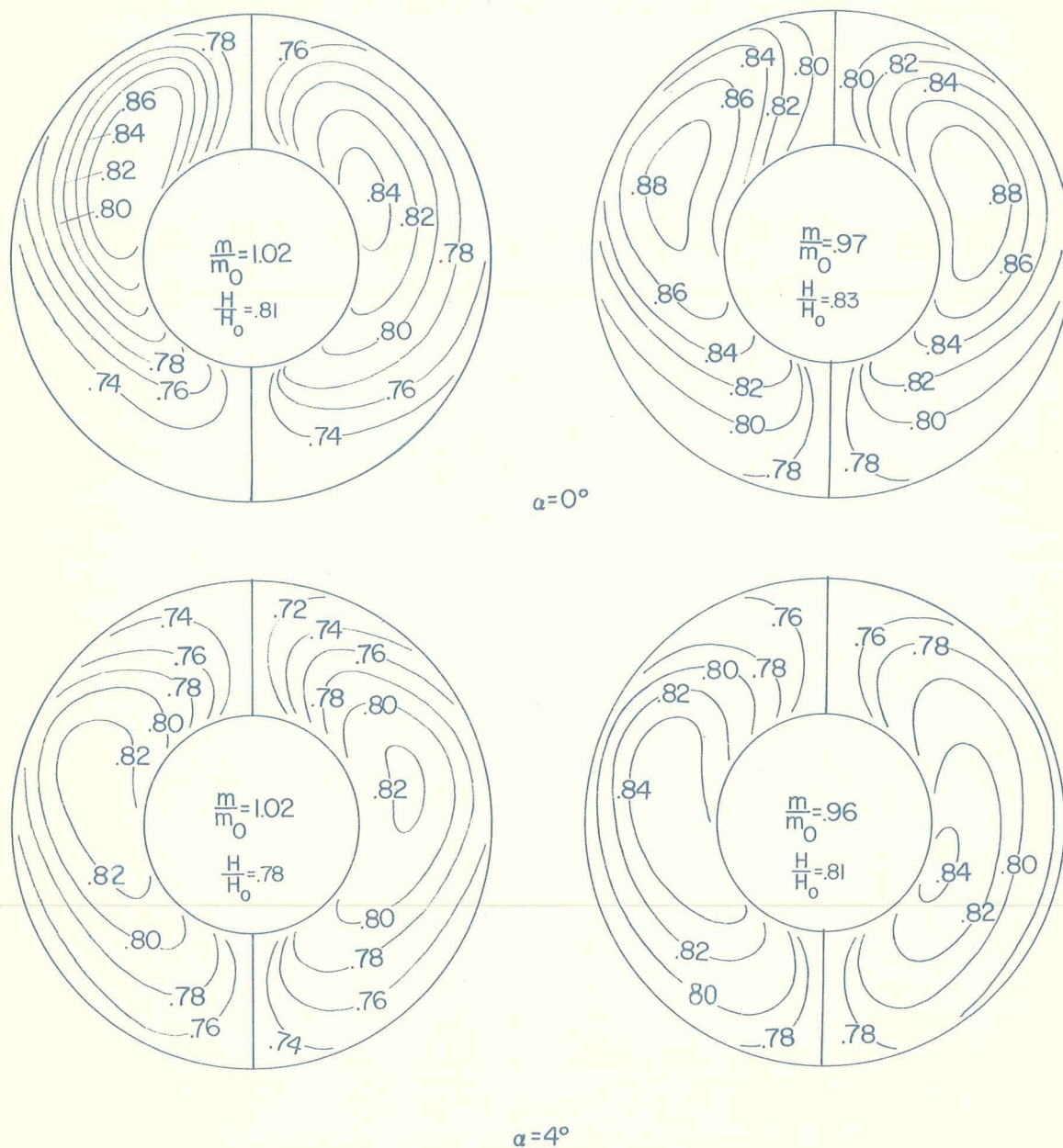


Figure 7.- Compressor-face total-pressure-recovery contours. Configuration 1; looking downstream; boundary-layer slots open 0.046 inch by 0.52 inch.

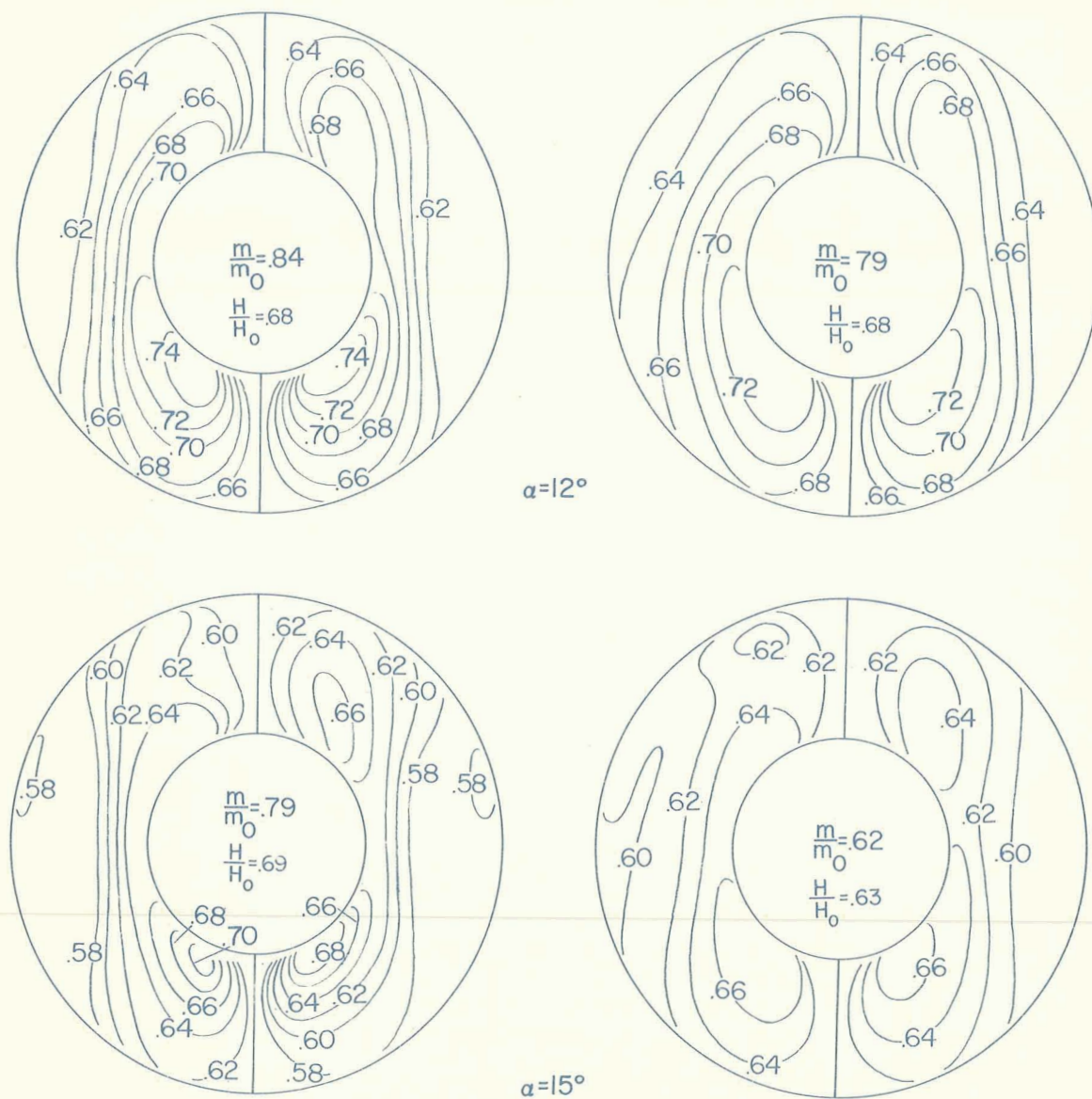


Figure 7.- Concluded.

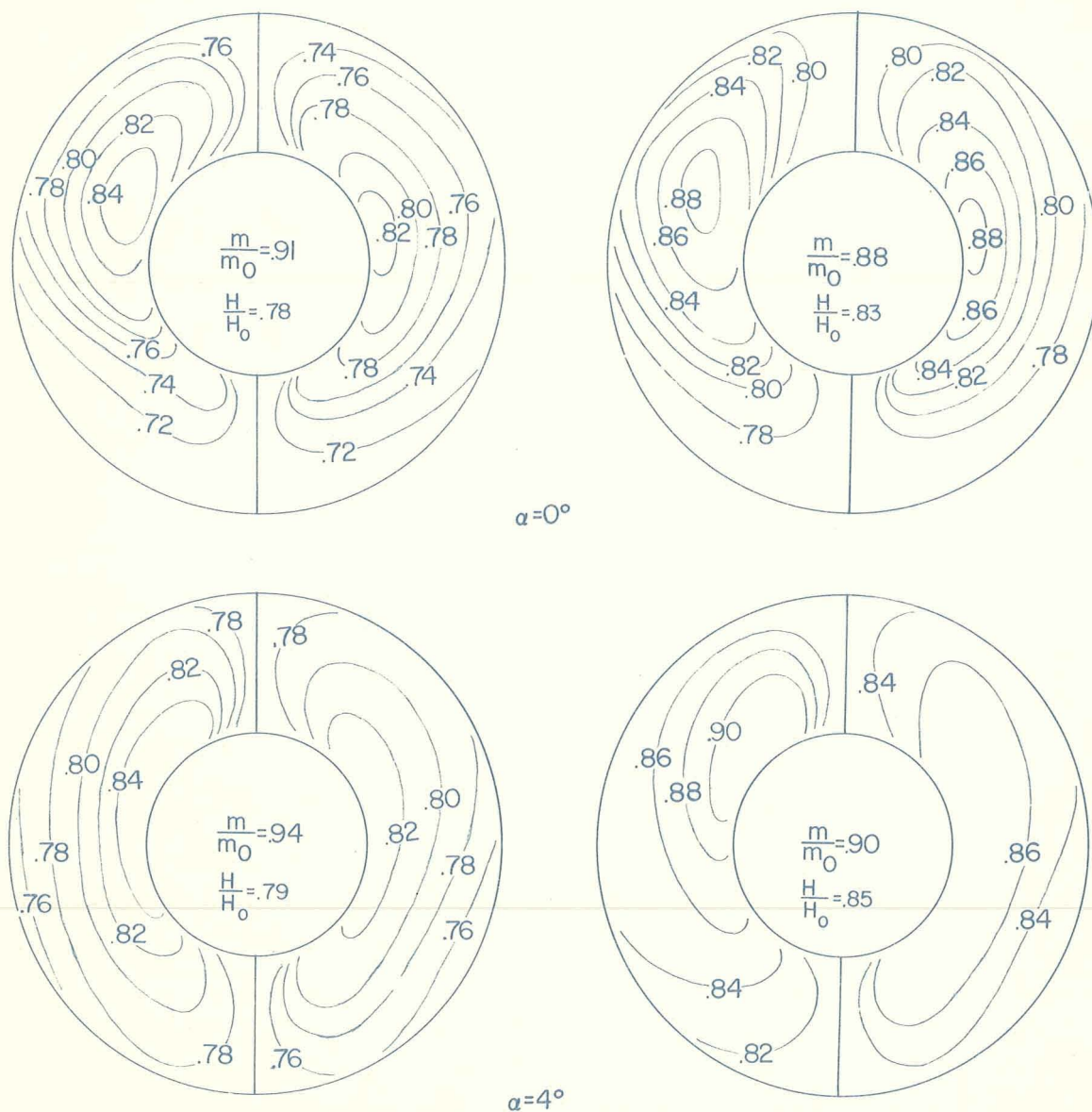


Figure 8.- Compressor-face total-pressure-recovery contours. Configuration 2; looking downstream; original boundary-layer plate; boundary-layer slots open 0.046 inch by 0.052 inch.

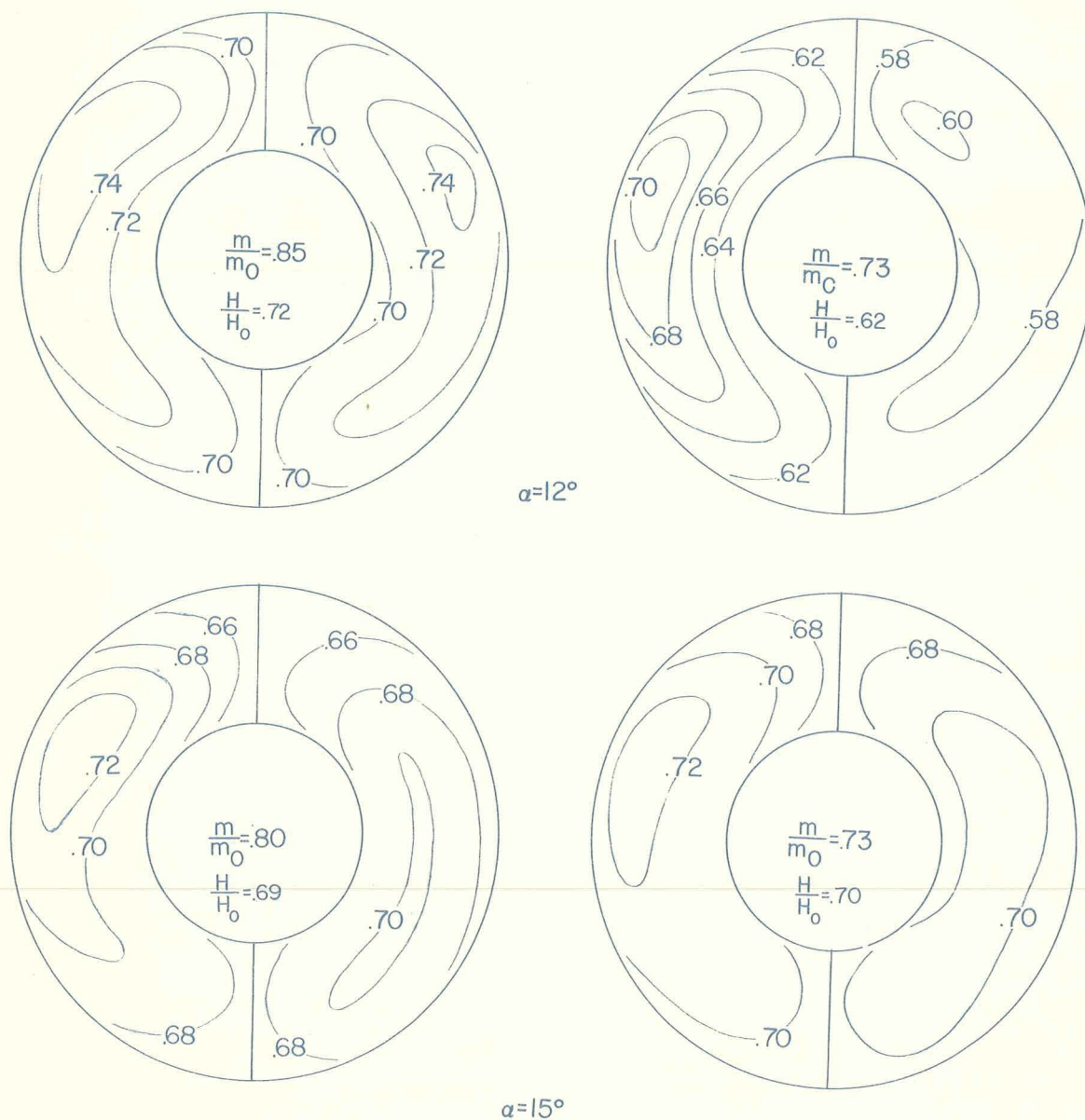


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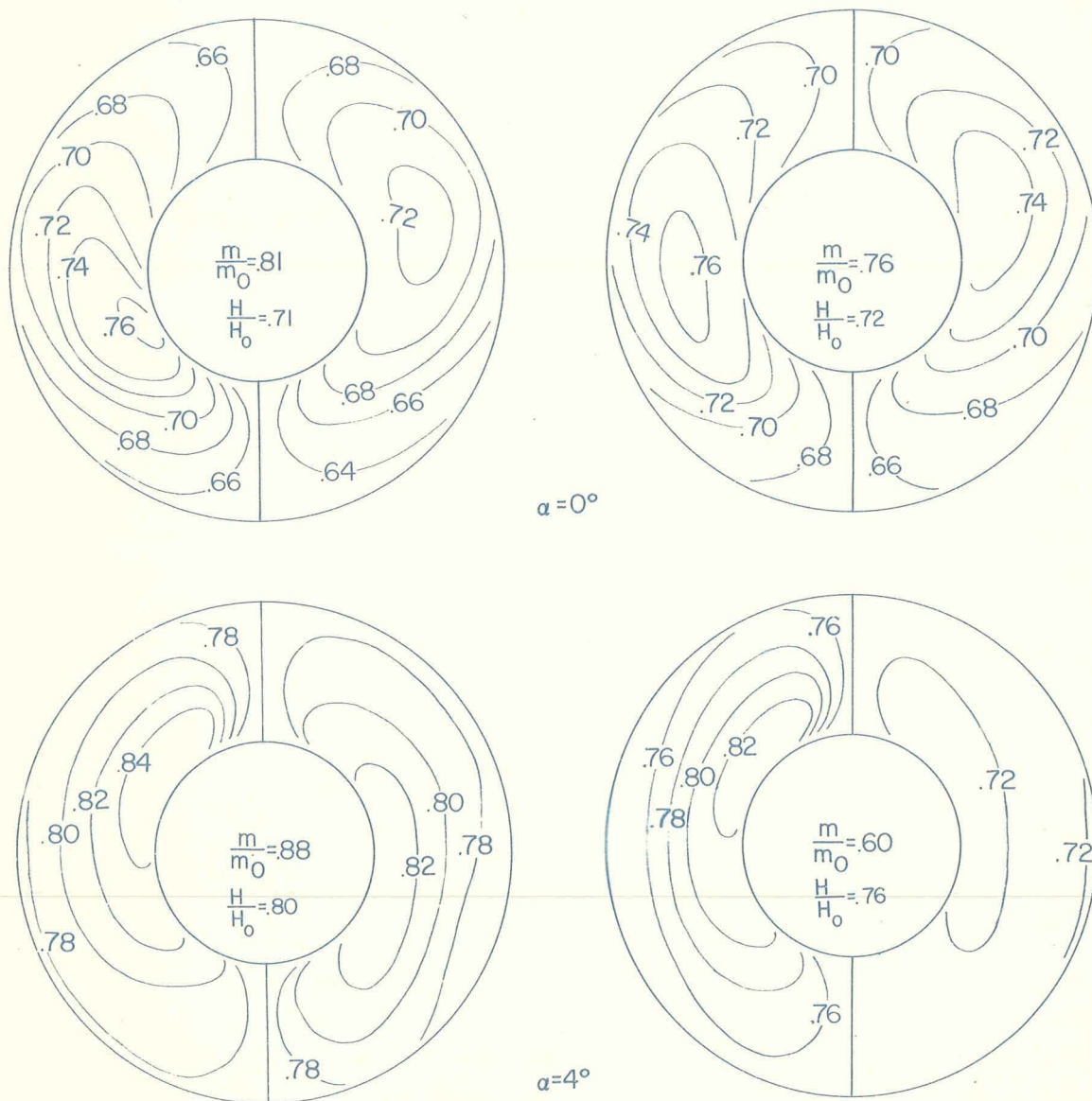


Figure 9.- Compressor-face total-pressure-recovery contours. Configuration 3; looking downstream; large boundary-layer plate; boundary-layer slots open 0.07 inch by 0.52 inch.

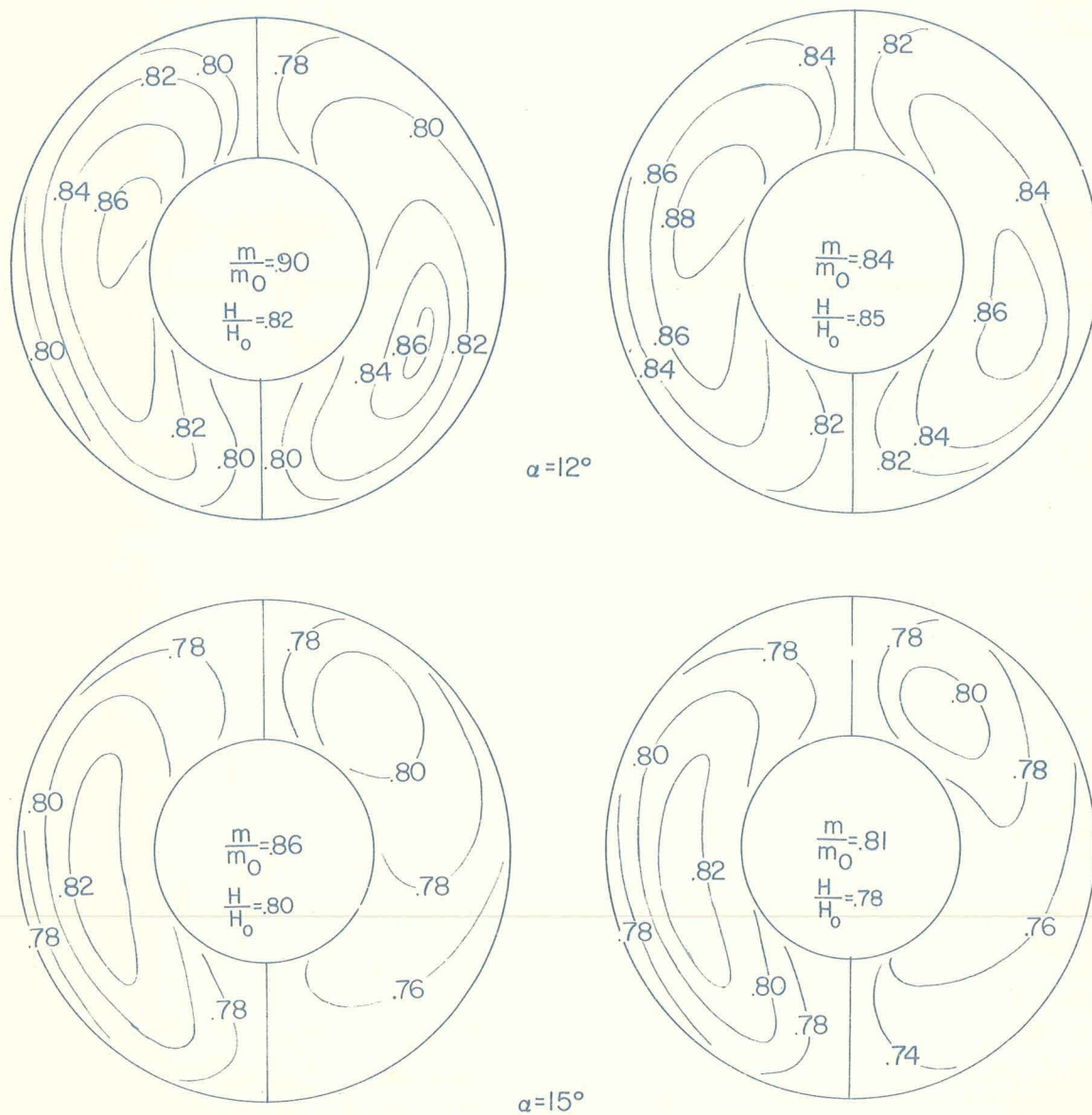


Figure 9.- Concluded.

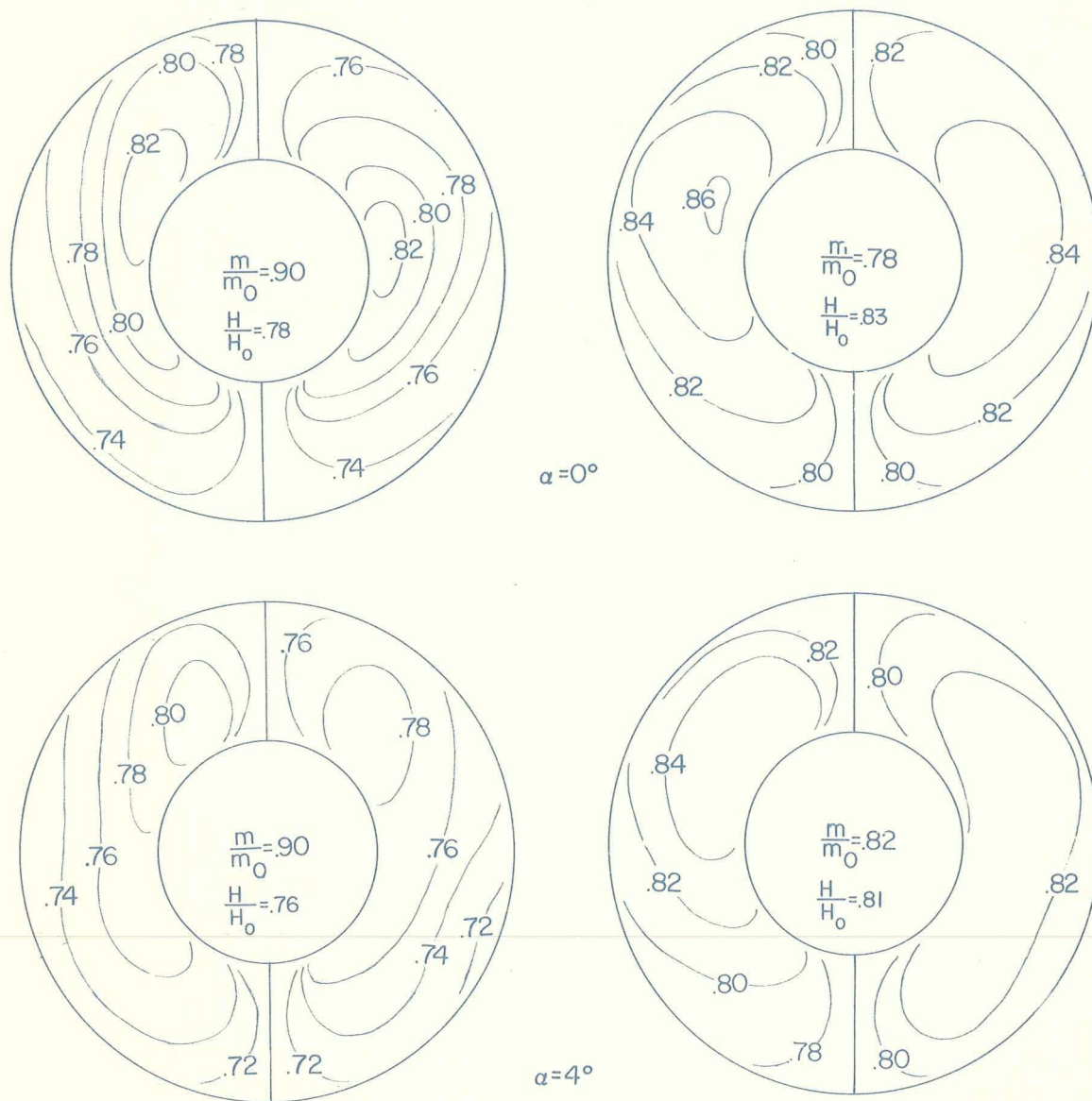


Figure 10.- Compressor-face total-pressure-recovery contours. Configuration 4; looking downstream; large boundary-layer plate; boundary-layer slots open 0.07 inch by 0.52 inch.

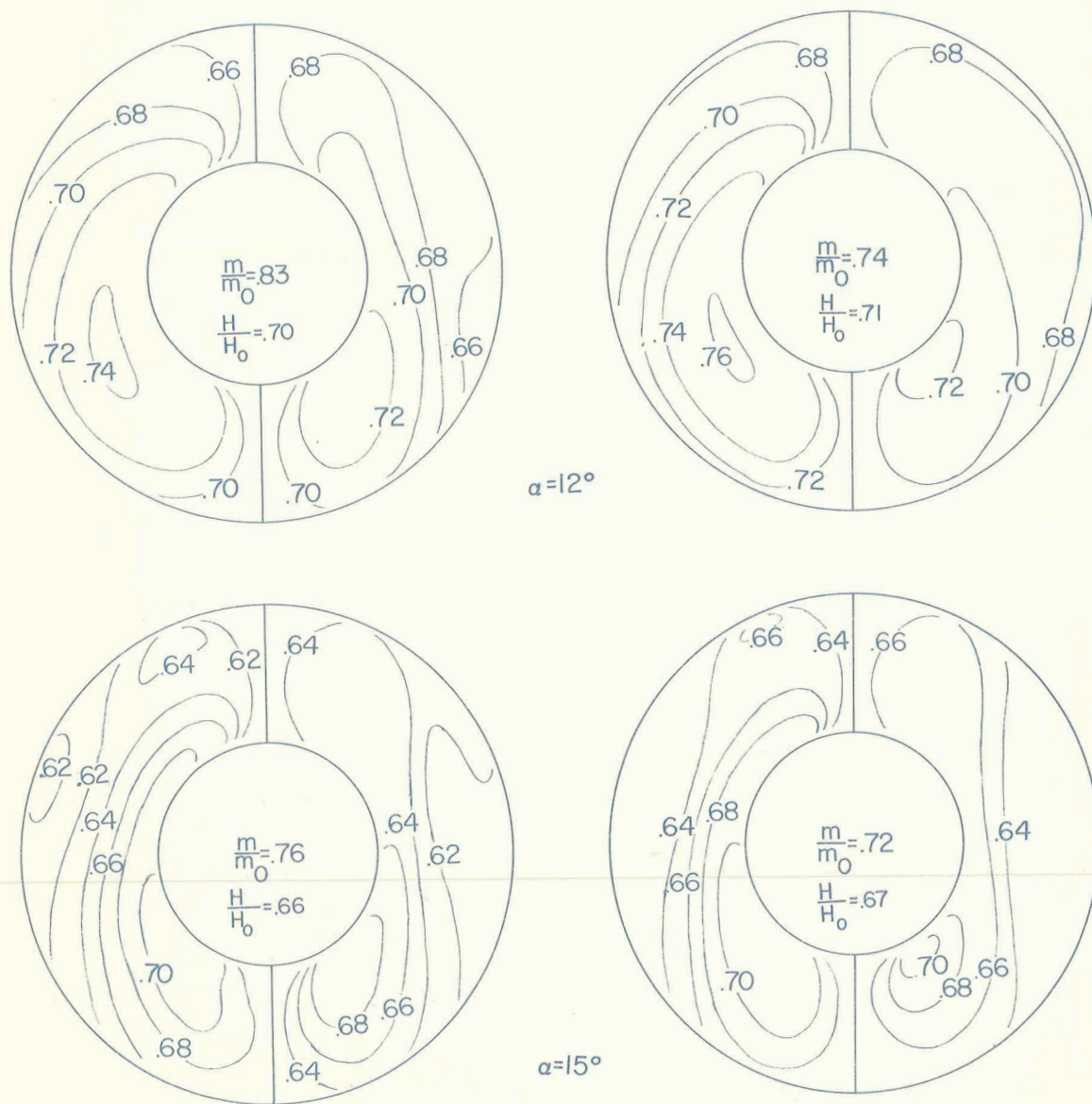
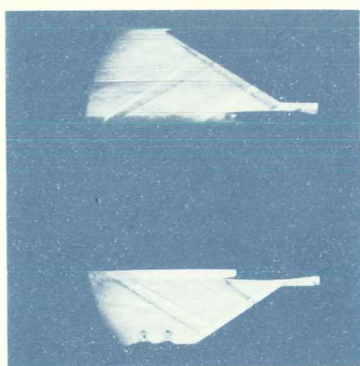
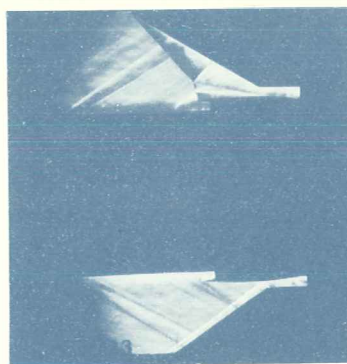


Figure 10.- Concluded.

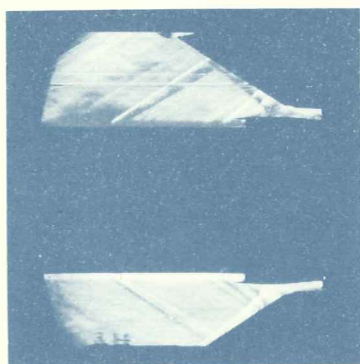


$$\frac{m}{m_0} = .97$$

$$\alpha = 0^\circ$$

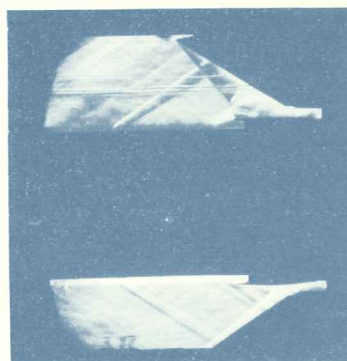


$$\frac{m}{m_0} = .60$$

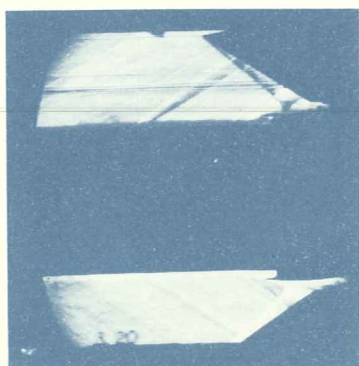


$$\frac{m}{m_0} = .79$$

$$\alpha = 12^\circ$$

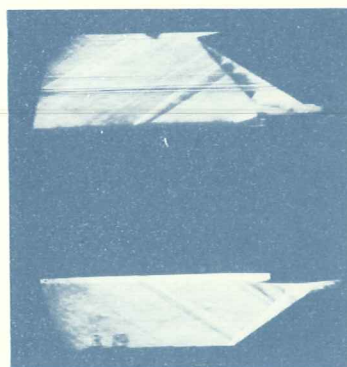


$$\frac{m}{m_0} = .59$$



$$\frac{m}{m_0} = .62$$

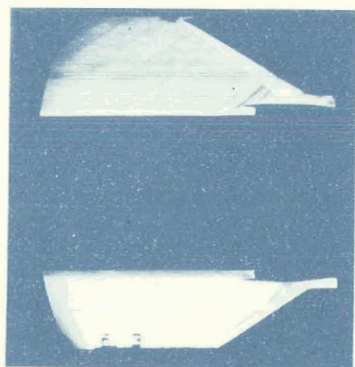
$$\alpha = 15^\circ$$



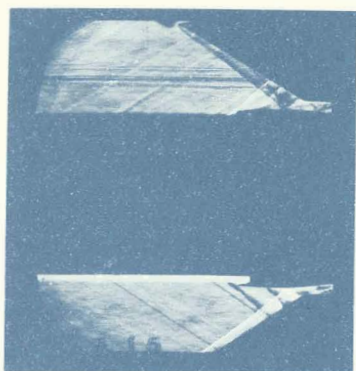
$$\frac{m}{m_0} = .56$$

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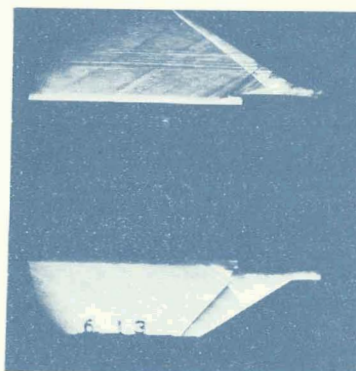
Figure 11.- Schlieren photographs of configuration 1. $M = 2.01$.

 $\alpha = 0^\circ$

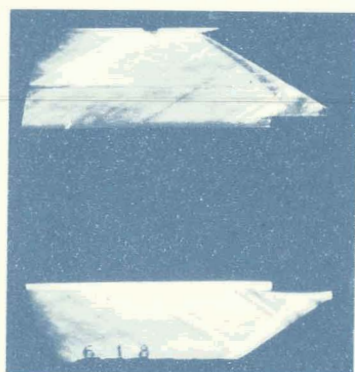
$$\frac{m}{m_0} = .90$$

 $\alpha = 12^\circ$

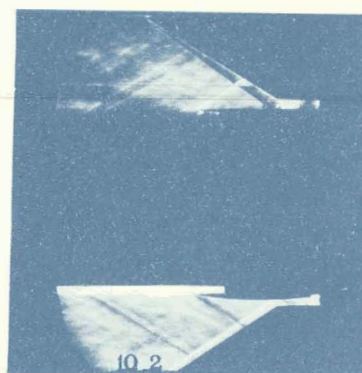
$$\frac{m}{m_0} = .84$$



$$\frac{m}{m_0} = .73$$



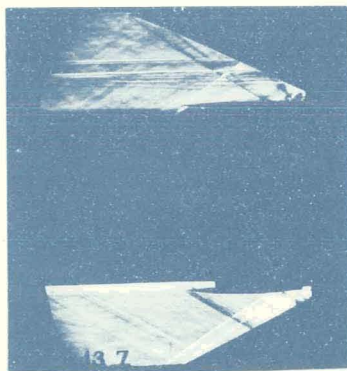
$$\alpha = 15^\circ \quad \frac{m}{m_0} = .73$$



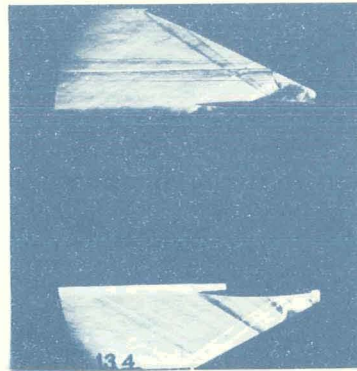
$$\alpha = 0^\circ \quad \frac{m}{m_0} = .87$$

(This photo with enlarged
boundary-layer floor plate)

Figure 12.- Schlieren photographs of configuration 2. $M = 2.01$. L-95909

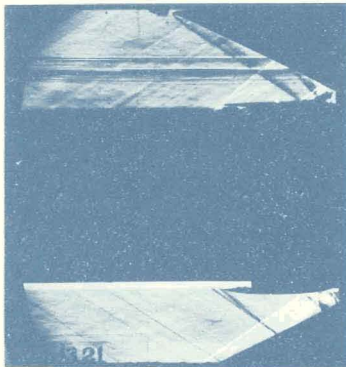


$$\frac{m}{m_0} = .90$$

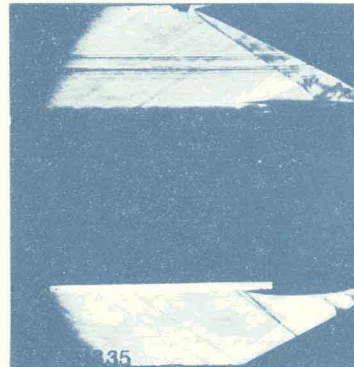


$$\alpha = 0^\circ$$

$$\frac{m}{m_0} = .78$$

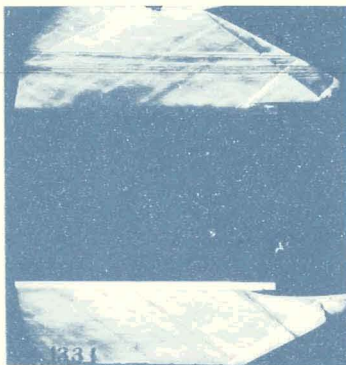


$$\frac{m}{m_0} = .82$$

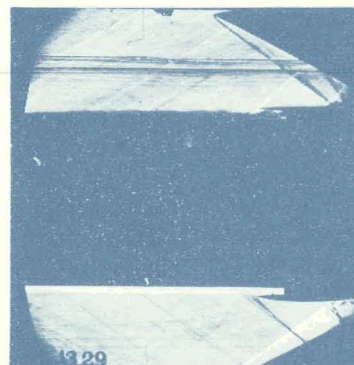


$$\alpha = 12^\circ$$

$$\frac{m}{m_0} = .72$$



$$\frac{m}{m_0} = .72$$



$$\alpha = 15^\circ$$

$$\frac{m}{m_0} = .51$$

Figure 13.- Schlieren photographs of configuration 4. $M = 2.01$. L-95910

PRELIMINARY WIND-TUNNEL INVESTIGATION OF THE PERFORMANCE
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VARIOUS ANGLES OF ATTACK AND A MACH NUMBER OF 2.01

By Walter L. Kouyoumjian

ABSTRACT

Several 1/13-scale Republic F-105 wing-root inlet configurations were tested at the Langley 4- by 4-foot supersonic pressure tunnel at a Mach number of 2.01 and a Reynolds number of approximately 3.4×10^6 per foot. The range of angle of attack was varied from -4° to 15° .

Inlet performance, compressor-face total-pressure contours, and schlieren photographs are presented.

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